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SUMMARY OF HEAT FLUX AND PRESSURE INSTRUMENTATION USED IN RECENT SATURN ROCKET EXHAUST TESTS

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ABSTRACT

Various types of instrumentation used in obtaining heat flux and pressure measurements in the exhausts of a number of Saturn rockets fired in the last two years are discussed. Analysis of experimental data obtained by various measuring techniques is an important factor in correlating predictions of pressure and heating from exhausts of liquid propellant main stage engines and solid propellant ullage and retro-motors used for stage separation purposes. The testing techniques at various facilities including type and location of instrumentation used and principal results obtained from the instrumentation are described.

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comparison of heating from Saturn solid propellant exhausts, describing briefly some of the experimental programs and discussing some preliminary results. The present report summarizes all of the experimental Saturn solid propellant tests and instrumentation used, as well as some of the most recent liquid (or gaseous) propellant firings. The number of firings of each type, test facility, type of measurement, approximate altitude, scale, and completion date are listed in table I.

II. DISCUSSION

A. Measurements in Saturn Solid Propellant Exhausts

The Saturn solid propellant firings discussed here involved the S-I, S-IB, S-IC, S-II, S-IVB, and Centaur retro and the S-II and S-IV (or S-IVB) ullage motors fired in the five test facilities listed in table I. All of the tests, with the exception of those at Cornell Aeronautical Laboratory, were full scale, and all of the tests except the firings at Rocketdyne, McGregor, Texas, were altitude firings. Since, in several cases, more than one type of motor was fired during a particular program, this section will be divided into the following groups of exhaust measurements: (1) S-IVB and S-II Ullage Exhaust Measurements at AEDC, (2) S-IVB Retro Exhaust Measurements at AEDC, (3) S-II Ullage Exhaust Measurements at OAL, (4) S-II Ullage Exhaust Measurements at Rocketdyne, (5) Centaur Retro Exhaust Measurements at MSFC, (6) S-IB and S-II Retro Exhaust Measurements at CAL, and (7) S-I, S-IB, S-IC Retro and S-II, S-IVB Ullage Exhaust Measurements at CAL.

1. S-IVB and S-II Ullage Exhaust Measurements at AEDC

These tests involved the firing of two S-IVB ullage (Thiokol TX-280) motors and two S-II ullage (Rocketdyne RS-U-501) motors in the spray chamber of the 300-foot deep, J-4 test cell at Arnold Engineering Development Center (AEDC) during the summer of 1965. The complete details of these tests, including instrumentation layout and test results are given by Rochelle in reference 3. Total and radiation heating and total pressure measurements in the plume were obtained by mounting two copper probes 23 feet above the motor exit plane. Radiation measurements were also obtained outside the plume by mounting two Hayes International radiometers, models FF-1 and AH-1, 6 inches and 10 feet, respectively, from the nozzle exit. The copper probes, shown in figure 1, built by Heat Technology Laboratory (HTL), were 2 1/2 inches in diameter and 10 inches long with a short extension on one end on which was mounted either a 0.1-inch thick teflon strip or a piece of uninsulated confined detonating fuse (CDF) assembly to qualitatively determine particle erosion effects. For the S-II ullage motor firings, one of the probes had a teflon-coated ordnance disconnect mounted on one end and, for the last firing, also had a silicone and sponge rubber insulated piece of CDF.

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SUMMARY

Various types of instrumentation used in obtaining heat flux and pressure measurements in the exhausts of a number of Saturn rockets fired in the last two years are discussed. Analysis of experimental data obtained by various measuring techniques is an important factor in correlating predictions of pressure and heating from exhausts of liquid propellant main stage engines and solid propellant ullage and retro-motors used for stage separation purposes. The testing techniques at various facilities including type and location of instrumentation used and principal results obtained from the instrumentation are described.

I. INTRODUCTION

The NASA Marshall Space Flight Center (MSFC) has the responsibility of determining the thermal environment to all portions of the Saturn vehicles caused by heating from the liquid propellant H-1, F-1, J-2, and RL-10 main stage engines and the solid propellant S-I, S-IB, S-IC, S-II, S-IV, S-IVB, and Centaur retro and the S-II, S-IV, and S-IVB ullage motors. Excessive heating and pressures from exhausts of main stage engines could damage the heat shields and could cause explosions in the base area. Also, excessive heating rates and pressures from solid propellant ullage and retro-motors could damage the nearby structure and vital components during the separation of stages.

For the past five years, calculations and tests sponsored by MSFC have been made in the Saturn base areas to determine radiation and convective heating from the main stage engines, and, for the past two years, in the stage separation regions, to determine radiative, convective, and particle impingement heating from the solid propellant motors. Several reports and papers have been published involving some of the calculations and tests, but none of these have summarized the most recent tests and the probes and optical instrumentation used in these motor firings. An oral presentation [1], made at the Fifth Annual Meeting of the Huntsville Section of the Instrument Society of America in April 1966, described some of the important measurements in Saturn solid propellant rocket exhausts up to that time. Rochelle [2] presented a theoretical and experimental

The copper probes had total and radiation calorimeters and a pressure tap mounted on the stagnation line. The calorimeters, built by HTL, were of the steady-state sensor or Gardon-type discussed in references 4 and 5. The total calorimeters had a thin, blackened (for maximum absorptivity) constantan disc of low thermal conductivity (13 BTU/ft hr °F) mounted to a copper heat sink of high thermal conductivity (233 BTU/ft hr °F). The sensitivity of the instruments was directly proportional to the square of the disc diameter and inversely proportional to its thickness. One copper wire connected to the copper mass and another butt-welded to the back side of the constantan disc formed a differential thermocouple whose equilibrium temperature was proportional to the absorbed heat flux from the rocket exhaust. The radiation calorimeters were also of the Gardon-gage-type with a 0.015-inch thick synthetic sapphire window around which a gaseous N_2 purge of about 150 psia was ejected. The pressure transducers were Hidyne Series 125 which required a 5V RMS, 20 KC excitation signal and registered a maximum output of 0.5 psi pressure differential. These transducers were mounted perpendicular to the total pressure tap in the probe to minimize particle impingement effects on the transducer diaphragm. The Hayes radiometers used outside the plume had a thermistor bolometer detector, 0.3 mm square, to determine total radiation emitted from the visible wavelengths out to about 15μ and had a field of view of 8.1 milliradians square.

Measured values of total heating rate on the probes for the S-II ullage motor were about 60-70 BTU/ft^{2}sec and radiation heating about 6-8 BTU/ft^{2}sec. The Hayes FF-1 radiometer measured about 20-22 BTU/ft^{2}sec near the exit plane and the AH-1 10 feet from the nozzle exit read less than 4 BTU/ft^{2}sec. Although steady-state readings obtained from the pressure transducer were about 0.25 psia, somewhat higher than predicted, it was believed that some of the solid Al_2O_3 particles were still impacting on the diaphragm. Total heating rates on the probe for the S-IVB ullage motor were about 60 BTU/ft^{2}sec, a relatively large value; however, because of the large area ratio ($A_e/A_t \sim 21$) of this motor, the first normal shock in the plume was found to occur a short distance in front of the probe, resulting in increased heating rates on the probe. The Hayes FF-1 radiometer near the exit plane read about 2-3 BTU/ft^{2}sec for the S-IVB ullage motor, and the AH-1 recorded less than 1 BTU/ft^{2}sec.

2. S-IVB Retro Exhaust Measurements at AEDC

These tests involved the impingement of the S-IVB retro motor (Thiokol TX-143) exhaust on a model of the Centaur stage in the spray chamber of the AEDC J-4 test cell as discussed by Muse [6]. Five motors were fired in this program, three at zero-degree cant angle and two canted at 11.5 degrees away from the vehicle. The object of these tests was to determine changes in reflectivity of the paint sample, laser beam attenuation through the plume, and solar cell panel erosion caused by impingement of Al_2O_3 particles, as well as surface pressure and heating caused by plume impingement along the Centaur panel. Total heating rates were obtained by Hy-Cal asymptotic calorimeters located at five positions along the panel, and radiative heating was obtained by an HTL calorimeter mounted outside the plume about 3.7 feet from the exit. Strain-gage-type pressure transducers of 1 psia maximum range, furnished by MSFC were mounted at seven locations along the panel. Total heating rates and pressures (averaged over time) are shown in figure 2 for a zero-degree cant angle firing. The theoretical correlation of these heating rates and pressures (along with those of other flat-plate-type tests) is scheduled to be presented shortly by Rochelle and Kooker [7].

3. S-II Ullage Exhaust Measurements at OAL

Infrared radiometer data were obtained from two S-II ullage motors fired at Ordnance Aerophysics Laboratory (OAL) in June 1965; spectrometer data were obtained from two S-II ullage motors fired at OAL in June 1966. The radiometer and spectrometer were mounted inside Altitude Cell No. 6, 5.73 inches from the exit plane. Two holes were cut in the diffuser walls (one on each side) for mounting adapters with a 0.065-inch thick sapphire window through which the detectors could receive the emitted radiation. The radiometer was the same Hayes FF-1 instrument used in the AEDC tests. Maximum heating rates recorded by the FF-1 were about 14-16 BTU/ft^2sec, somewhat less than the AEDC measurements, but it was believed that considerable Al_2O_3 and soot deposits on the window somewhat reduced transmissivity. The spectrometer was a Block Engineering Model BD-1 with a scan rate of 1-4 scans/sec which had four detectors (one channel per detector and two orders to each channel): (1) a 1P28 photomultiplier tube for 0.22-0.33μ and 0.33-0.66μ, (2) lead sulfide detector for 0.66-1μ and 1-2μ, (3) lead selenide detector for 1.8-2.3μ and 2.3-4.6μ, and (4) a thermistor bolometer for 4-6μ and 6-12μ. Since the spectrometer measurements were limited to only one channel (two orders) per firing for two firings, the complete spectrum was not surveyed. However, good spectral data were obtained in the 2.7μ H_2O and CO_2 band (0.7-0.8 watts/$cm^2\mu$ sterad), the 1.8-2.3 Al_2O_3 and Fe_2O_3 continuum region (0.5-0.7 watts/$cm^2\mu$ sterad), and the HCl band (0.2-0.25 watts/$cm^2\mu$ sterad).

4. S-II Ullage Exhaust Measurements at Rocketdyne

Two hemispherical probes, three inches in diameter and five inches long (one copper and one teflon), were mounted 20 inches from the exit plane in two S-II ullage motor firings at Rocketdyne, McGregor, Texas, in April 1966. These probes, which were manufactured by HTL, are described and results of the tests are reported by Datis and Fowler [8]. Figure 3 shows the installation and mounting arrangement for the copper probe above an S-II ullage motor before firing. The copper probe contained a 0.5-inch diameter copper slug in which were imbedded nine chromel-alumel thermocouples which, together with five additional chromel-alumel thermocouples mounted radially, were used to obtain the probe temperature distribution as a function of time, and consequently the stagnation point heating. From this measured temperature distribution, it was calculated that stagnation point heating rates reached a level of 3000-3200 BTU/ft^{2}sec after the chamber pressure reached steady-state conditions.

Ablation rates were obtained on the teflon probe by positioning 12 thin copper-constantan thermocouple wires in a 0.5-inch diameter teflon slug at various distances back of the stagnation point. Output was recorded on analog as well as digital recorders. It was possible to obtain within a few milliseconds the exact time at which each thermocouple wire was burned through. Therefore, the ablation rates all of the order of 0.28-0.33 in/sec (see fig. 4) were obtained quite accurately as a function of time. The dotted line, representing the ablation depths calculated by the program described in reference 9, shows very good agreement with the data.

5. Centaur Retro Exhaust Measurements at MSFC

Ten Centaur retro-motors were fired in Altitude Cell 112 at MSFC's Test Laboratory from March to June 1966. The following five motors (two of each type) were fired during the program: (1) Rocket Power, Inc. (RPI) 13 percent aluminum BAP-8 propellant, (2) Thiokol polysulfide TX-3, (3) United Technology Center (UTC) UTX7757, (4) RPI two percent aluminum PAP-8 propellant, and (5) Atlantic Research Corporation (ARC) Arcocel 268 nitrocellulose. The details of this program, including instrumentation and test results, are reported by Rochelle [10]. All of the plumes of the motors impinged upon a flat plate mounted 12 inches from the axis of each motor whose nozzle exit was positioned 5.5 inches from the end of the plate as seen in figure 5. Although laser reflectivity, laser velocimeter, and paint sample degradation experiments were performed, as well as heating and pressure measurements, these are not discussed here. A radiation calorimeter and nine total calorimeters (of the Gardon type) manufactured by HTL were mounted on the centerline of the 36-inch by 96-inch flat plate which was placed vertically in the 8-foot by 37-foot test cell. Nine pressure transducers (Statham Model 80 differential and Consolidated Electrodynamics Corporation type 4-353-0100 absolute) were also mounted along the plate centerline. A Hayes FF-1 radiometer was mounted outside the plumes about 5 inches from the exit plane and about 31 inches from the axis of the motors.

Experimental values of total heating and pressure on the flat plate centerline are shown in figures 6 and 7, respectively. For most of the firings, the peak in heating rate and pressure appears to be roughly proportional to motor chamber pressure. An additional peak in heating rate and pressure for the ARC motor is evident far downstream because of the normal shock formed on the plate due to the motor's large expansion ratio which forced the lip shock to come down faster than any of the other motors. The Hayes FF-1 radiometer mounted outside the plume measured average radiation heating rates of the level of 6 BTU/ft^2sec or less for all of the motors. The theoretical correlation of much of the flat-plate data in figures 6 and 7 will be discussed by Rochelle and Kooker [7].

6. S-IB and S-II Retro Exhaust Measurements at CAL

These solid propellant motor tests were performed at Cornell Aeronautical Laboratory (CAL) during 1966 and early 1967. Two impingement configurations were tested: (1) the S-IB retro-motor exhaust impinging on the S-IB interstage (region between nozzle exit and S-IB/S-IVB separation plane) and on the S-IVB thrust structure at 200,000 feet and (2) the S-II retro-motor exhaust impinging on the S-II interstage (region between nozzle exit and S-II/S-IVB separation plane) and on the S-IVB thrust structure at 391,000 feet and 200,000 feet. Since 391,000 feet was the highest altitude obtainable in the test cell, data were extrapolated to 600,000 feet, the approximate altitude at actual S-II/S-IVB stage separation. The preliminary test plan for this investigation was given by Rochelle [11] and a later modified version was described by Dennis [12]. The tests were all performed in a 10-foot by 28-foot altitude cell in which were mounted the 1/10-scale models and a solid propellant combustor capable of duplicating the chamber pressure, propellant, and expansion ratio of the full scale motor. This combustor, discussed by Hendershot [13], is composed of basically three parts: (1) combustion chamber in which thin sheets of propellant are glued to the surfaces of a 10-point star propellant holder, (2) an exhaust nozzle, and (3) venting diaphragms. Within 2 to 3 milliseconds the propellant surfaces were ignited uniformly by a spark plug in an oxygen-rich (O/F = 20) H_2-O_2 mixture, and after the nozzle diaphragm ruptured at the required chamber pressure (P_c = 1770 psia), a period of 15-20 milliseconds of steady-state testing time existed for each run.

Pressure and heating data were obtained on the 1/10-scale models of the S-II/S-IVB configuration shown in figure 8 and on the S-IB/S-IVB configuration for separation distances of 0 to 60 inches. On the interstage, pressure and thin film heat transfer gages (described by Bogdan in references 14 and 15) were placed on the top of the stringer underneath the axis of the motor and at angles, θ, of 4.82, 11.25, 17.68, and 27.33 degrees from this position. Gages were also placed on portions of the J-2 nozzle and on various components such as models of the helium sphere, hydrogen feed line, hydraulic pump, hydraulic accumulator, propellant chill-down return line, and electrical connect panel of the S-IVB stage. To check scale effects on stagnation heating, various sphere scaling tests were run in which the overall scale of the nozzle, the sphere size and distance of sphere from nozzle exit were varied, but the motor chamber pressure, propellant, nozzle expansion ratio, and altitude were unchanged.

The total heating gages (shown in figure 8) mounted at 27.33 degrees from the plume axis were composed of a 0.1μ thick platinum strip which had planform dimensions of approximately 0.03 inches by 0.25 inches and which was fused on the front surface of a pyrex glass substrate. A very thin ($\sim$ 4 micro-inches thick) evaporated coating of MgF_2 was placed on the platinum film to provide electrical insulation from the ions in the exhaust. Radiation heating rates were measured near the nozzle exit plane and on the S-IVB stage base with dual-element thin film gages which had two platinum strips mounted perpendicular to each other (total heating on front side and radiation heating on rear side). The substrate used for these gages was quartz because of its better transmissibility for radiation measurements. Static pressures on the interstage, S-IVB thrust structure, and J-2 engine were measured by CAL-developed piezoelectric crystals with high electrical sensitivity (1500 mv/psi) but which were relatively insensitive to acceleration effects. Total pressures on the S-IB interstage were measured at distances of 2.23 inches, 4.72 inches, 9.22 inches, and 10.74 inches from the nozzle exit and on the S-II interstage at distances of 2.14 inches and 5.49 inches from the nozzle exit by means of a 5-probe nozzle rake with probes 0.25 inches apart.

The preliminary results of these tests have been reported by Dennis in reference 16. Figure 9 shows heating rates and pressures on the S-II interstage ($\theta = 0°$) at 391,000 feet showing a maximum near the exit plane and then decreasing sharply to a point downstream where it is believed the reflected shocks off adjacent stringers tended to somewhat increase surface pressures and heating rates. Some of the gages on the S-IVB thrust structure nearest the projection of the nozzle centerline recorded heating rates above 100 BTU/ft^2sec. Because a large amount of impacted particles (Fe_2O_3 and carbon) was found in this region, it was believed that most of the heat transferred there was caused by the large transfer of the thermal and kinetic energy from the particles to the surface. The radiation gage near the nozzle exit read only about 1.5 BTU/ft^2sec, a value quite low for solid propellant exhausts. The total heating rates along the interstage compared well with the in-flight data and preliminary calculations discussed in reference 17. Total pressure probe data, which compared well with the theoretical predictions, have been discussed in more detail in reference 7, along with the pressure and heat transfer calculations for the S-IB and S-II interstages.

7. S-I, S-IB, S-IC Retro, and S-II, S-IVB Ullage Exhaust Measurements at CAL

These tests, conducted in the CAL test cell, involved a comparison of the spectral profile of the exhausts of the S-I, S-IB, and S-IC retro-motors and the S-II and S-IVB ullage motors. A Warner-Swasey Model 501 infrared spectrometer was the main instrument used; three PbS detectors and a photomultiplier tube were used to further monitor the radiation. The Model 501 spectrometer, with a field of view of 1.25 inches by 0.125 inches, was centered at 0.125 inch from the exit plane, and was viewed through a 4-inch diameter sapphire window at a distance of 63 inches from the motor axis (outside the cell). The radiant intensity was obtained from 1.7 to 5.0μ with approximately four scans per firing (1 scan per sec with 0.25-second turn-around time). A 1P28 photomultiplier tube monitored the radiation from 0.23 to 0.6μ, and one of the PbS detectors recorded radiation from the visible to 3μ, while the other two measured at 2.2 and 2.7μ. Test data described by McCaa [18] show generally that the peak intensity levels varied directly with the product of the theoretical exit plane temperature and pressure, and were the highest for the S-IB retro with a peak intensity in the 2.7μ H_2O and CO_2 band of about 2.6 watts/cm^2μ sterad and in the 4.3μ CO_2 band of about 5.5 watts/cm^3μ sterad.

B. Recent Measurements in Saturn Liquid and Gaseous Propellant Exhausts

In the last two years several documents have been published summarizing base heating experimental measurements and in-flight data from Saturn liquid propellant engine firings and gaseous propellant firings using the CAL short-duration combustor which simulated the liquid propellant firings. Hendershot [19], Hopson and McAnelly [20], and Dearing [21] have all discussed the effects of base heating caused by the exhausts of clustered liquid and gaseous propellant engines. Wilson [22,23] discussed the clustered engine base heating problem placing special emphasis on the use of the CAL short-duration combustor. Payne and Jones]24] and Jones [25] discussed the base thermal environment on the Saturn I vehicles, describing both inflight and scale model data. McEntire, Mullen, and Fowler [26] summarized all of the S-IC scale model tests using the CAL short-duration combustor with gaseous O_2 and ethylene to simulate LOX and RP-1, respectively. Reference 26 included all the S-IC external flow data obtained in the CAL 8-foot by 8-foot transonic wind tunnel, the NASA/Lewis 8-foot by 6-foot transonic wind tunnel, the NASA/Lewis 10-foot by 10-foot supersonic wind tunnel, and the CAL Mach 2 high altitude chamber. Several full scale J-2 and F-1 engines and one S-IC stage (5 F-1's) have been fired at MSFC, and several scale model F-1 and J-2 engines have been fired at CAL recently in which radiation measurements have been obtained. Complete results from these tests have not yet been documented.

The present report describes heating and pressure instrumentation and measurements from five Saturn liquid and gaseous propellant tests. Of this group of tests, only one has actually been completed, two are in progress, and two are in the planning stage. These tests pertain to both base heating and pressure measurements and to heating and pressure measurements on various objects placed inside the plumes. The instrumentation and measurements discussed here pertain to the following: (1) F-1 tests at MSFC, (2) S-IC (5 F-1's) tests at MSFC, (3) J-2 tests at MSFC, (4) S-IC (5 F-1's) tests at CAL, and (5) J-2 tests at AEDC.

1. F-1 Exhaust Measurements at MSFC

This test program, which involved nearly a hundred firings of a 1/20-scale F-1 LOX/RP-1 engine performed at MSFC's Test Laboratory, is described in the final data report by Patrick [27]. Radiation measurements were obtained by means of eleven HTL narrow-view angle (4.6°) radiation calorimeters mounted parallel to the engine axis and 30 inches to the side of the axis. The first calorimeter was 3 1/2 inches from the exit plane, the next seven were 7 inches apart, and the next three were 14 inches apart. Other wide-view angle calorimeters (90-135°), built by HTL and Hy-Cal, were mounted 11 inches behind the nozzle exit at distances of 7, 8, 11 1/2, and 18 1/2 inches from the nozzle axis. Chrysler total and radiation calorimeters (purged with gaseous N_2) were mounted inside the plume on the axis at values of X/D_e of 12, 15, 20, 25, and 30 from the nozzle exit. Total and static pressure and temperature measurements were obtained in the plume by means of three-pronged rakes with 2 1/3 inches spacing between prongs. These were mounted in the plume at values of X/D_e of 12, 15, 20, 25, 30, 40, and 50. Two types of pressure probes (which used Wianco strain-gage transducers) were used on the rakes: (1) stainless steel with ablative carbon on the outside and (2) solid copper with 1/8-inch stainless steel inner tubing. The temperature probes had tungsten 10 percent rhenium wires mounted in a stainless steel tube which also had porcelain insulating material on the inside. A steel shield, which protected all the gages and probes before ignition, was dropped hydraulically less than a second after ignition, and then was moved up again to cover the gages during engine shutdown.

Discussions of the narrow-view angle radiometer data for these tests have been presented by Murphy, et al. [28] and Hiserodt [29]. Various peaks in radiation heating along the axis were experienced because of the strong normal shocks occurring fairly close to the nozzle exit. The maximum radiation heating rate of 107 BTU/ft^{2}sec was recorded at an X/D_e of 6.5 (behind the third normal shock). Radiation measurements on the calorimeters inside the plume varied from about 155 BTU/ft^2-sec at an X/D_e of 25. Total heating measurements in the plume varied from about 700 BTU/ft^{2}sec at an X/D_e of 12 to about 60 BTU/ft^{2}sec at an

X/D_e of 30. The total pressure measurements are shown in figure 10, together with calculations based on the viscous shear layer discussed by McGimsey [30]. It may be seen that very good agreement existed, especially far down stream. The peaks in the inviscid plume, calculated by methods discussed in references 31-33, represent theoretical total pressure behind the normal shocks. These peaks are believed to be valid to only a few diameters down stream because of viscous mixing taking place throughout the plume in this region.

2. S-IC (5 F-1's) Exhaust Measurements at MSFC

This test program involves the firing of a 1/58-scale model of the S-IC stage with the exhausts impinging on models of the Saturn V launch umbilical tower (LUT) and on the flame trench as the model dynamically simulates various lift-off trajectories. This test is in progress, and a preliminary listing of heat flux and pressure gages has been discussed by Rochelle [34]. Reference 34 also describes a preliminary theoretical analysis of heating and pressure to the LUT swing arms, the results of which will be used in an attempt to correlate with the experimental data. Figure 11 shows the test setup with the lower portion of the LUT in place showing the F-1 nozzles (soon to be replaced with water-cooled ones), flame trench, and location of some of the instrumentation. It is planned to mount scale models of four 120-inch (and later, 156-inch) solid propellant motors on the S-IC stage during the summer of 1967 and examine particle impingement effects on the surfaces in an effort to determine whether Launch Complex 39 at Cape Kennedy will have to be extensively refurbished or whether a new test stand will have to be built to handle effects of the strap-ons.

Gardon-gage calorimeters built by HTL are being used with maximum total heating rates of up to 1000 BTU/ft^2sec (water-cooled) where the plumes impinge on the top of the LUT deck. Several Statham strain-gage pressure transducers are mounted in each swing arm, as well as on the LUT deck and top of the LUT. Two types of gas total temperature probes are being used: (1) an MSFC-built chromel-alumel exposed-bead type used on the swing arms and expected to measure temperatures up to about 2500°R and (2) tungsten 5 percent rhenium/tungsten 26 percent rhenium used on the LUT deck and top of the LUT and expected to measure temperatures up to about 4500°R. Preliminary results indicate that total heating rates on the LUT deck were about 600 BTU/ft^2sec, radiation heating about 100 BTU/ft^2sec, and total pressures about 11 psia. Total heating on swing arm number 2 was only about 45 BTU/ft^2sec and radiation heating about 22 BTU/ft^2sec, even for the worst possible case of maximum drift trajectory.

3. J-2 Exhaust Measurements at MSFC

This program involves the firing of a 1/12.25-scale model of the J-2 engine using LOX and gaseous H_2 as propellants at sea level conditions at the MSFC Test Laboratory. The test setup in figure 12 shows a total calorimeter probe mounted near the exit of the model J-2 engine. The current test plan calls for mounting total pressure probes, total and radiation calorimeter probes and tektite ablation sample probes in the exhaust at distances up to 14-16 inches from the exit plane. All probes (and instruments) will be shielded from ignition and cut-off spikes in the same manner as for the scale model F-1 MSFC firings. HTL and Hy-Cal total calorimeters, mounted in hemispherical and blunt-faced cylindrical configurations, respectively, are expected to measure total heating rates in the neighborhood of 1500 BTU/ft^{2}sec. Radiation heating will be measured by means of both Hy-Cal and Chrysler-built radiation calorimeters capable of upper limits of around 250 BTU/ft^{2}sec and purged with gaseous N_2. A stainless steel pressure probe, fitted with a Wianco pressure transducer ranged up to 150 psia, will be used to obtain total pressures in the exhaust. Experimental values of stagnation point convective heating (found by subtracting total measurements from radiation measurements) will be correlated using the theoretical methods of Golden [35] and Fay and Riddell [36].

4. S-IC (Five F-1 Engines) Exhaust Measurements at CAL

This program involves testing a 1/45-scale model of the S-IC stage with five F-1 engines and with model 120-inch solid propellant strap-ons at CAL. It is currently planned to conduct the program in three phases: (1) sea level LUT tests at CAL (without external flow), (2) Mach 2 external flow tests in CAL short-duration external flow facility, and (3) transonic (M_∞ = 0.6, 0.8, 1.0, and 1.2) wind tunnel tests in the CAL 8-foot by 8-foot transonic wind tunnel. The sea level tests will complement the 1/58-scale model tests at MSFC described previously. It is planned to begin testing July 1967 on phase 1 and complete the program by November 1967 as stated in the preliminary test plan written by Hendershot and Dennis [37]. Since the CAL short-duration combustor will be used, the LOX and RP-1 propellants will be simulated by gaseous O_2 and ethylene, respectively, and the solids will be simulated by use of 4 solid propellant combustors. The turbine exhaust will be simulated by injecting hydrogen into the F-1 nozzle from the turbine exhaust manifold. The fast-response pressure gage and thin-film heat transfer gage, used on the CAL solid propellant tests previously discussed, will be used on the base. However, because of the highly erosive properties of the Al_2O_3 particles from the 120-inch strap-ons, it is planned to use Hidyne slug-type calorimeters on the LUT deck for phase 1 of the tests. These calorimeters will have a thin ($\sim$ 0.10 inch thick) aluminum disc as the calorimetric mass with a platinum resistance thermometer mounted on the back to measure disc temperature. It is also planned to use a Warner-Swasey fast-response spectrometer to measure infrared radiation from both the F-1 and solid propellant exhausts.

5. J-2 Exhaust Measurements at AEDC

Although these measurements in the J-4 test cell at AEDC have been planned for over a year, they have not yet been made because of the higher priority of the J-2 re-start firings now under way which must be finished before the first Saturn V launch from Cape Kennedy. However, it is presently planned to mount two water-cooled probes (1 in each firing), manufactured by HTL, approximately 90 inches down stream of the J-2 nozzle exit inside the 13 1/2-foot diffuser. The probes, which will have a cylindrical nose piece about 3 inches in diameter, will contain an HTL Gardon-gage type calorimeter ranged up to 1000 BTU/ft^{2}sec and a radiation calorimeter of 100-150 BTU/ft^{2}sec maximum range. It is also planned to have a Statham strain-gage pressure transducer of 150 psia range on the stagnation line to measure total pressure and another Statham strain-gage transducer of about 5 psia at the side of the probe to measure static pressure. A Hayes FF-1 radiometer will also be used with filters to detect the H_2O emission in the 1.4, 1.8, 2.7, and 6.3μ wavelength intervals. This radiometer will be mounted on top of the diffuser insert to view through the exhaust at the exit plane. Stagnation line convective heating data are expected to be correlated by the methods described in References 35 and 36, and radiation heating data by the methods discussed in References 38 and 39.

III. CONCLUSIONS

This report has summarized heat flux and pressure instrumentation used in exhausts of recent Saturn solid, liquid, and gaseous propellant firings. Measurements in all of the Saturn solid propellant firings and measurements in the five most recent liquid (and gaseous) firings were described. In general, three types of measurements were obtained: (1) flat-plate or curved panel, in which the exhaust impinged parallel or at a small angle to the motor axis, (2) stagnation, in which spherical or cylindrical probes were completely immersed in the exhaust plume and (3) optical, in which the measurement was sensed from outside the plume. Many of these heating and pressure data have been and will be used as design criteria for the Saturn base region (liquid, gaseous, and solid strap-on propellant firings) and for stage separation regions (solid propellant ullage and retro firings). Convective flat-plate heating and pressure data are being correlated by theoretical methods discussed in reference 7, convective stagnation point heating data by methods described in references 35 and 36, gaseous radiation by methods mentioned in references 38 and 39, and particle radiation by methods outlined in references 39 and 40. It is possible that additional experimental measurements may be obtained in the near future from exhausts of engines used in the Apollo applications and Voyager programs. A fairly comprehensive review of the types of instrumentation which NASA/MSFC engineers feel are essential to adequately determine thermal and pressure environments in Saturn rocket exhausts has been presented.

Table I
Saturn Rocket Exhaust Tests

Motor	No. Firings	Test Facility[1]	Type Measurement[2]	Approximate Altitude (ft)	Scale	Completion Date
Solid Propellant Motor Exhaust Tests						
S-IVB Ullage	2	AEDC	S; P,TC,RC,R	110,000	Full	Aug. 1965
S-II Ullage	2	AEDC	S; P,T,TC,RC,R	120,000	Full	Aug. 1965
S-IVB Retro	5	AEDC	FP; P,TC,RC	115,000	Full	Nov. 1965
S-II Ullage	4	CAL	R,SP	140,000	Full	June 1966
S-II Ullage	2	RD./McG.	S; T,TC,A	Sea level	Full	April 1966
Centaur Retro	10	MSFC	FP; P,TC,RC,R	120,000	Full	May 1966
S-IB Retro	~ 50	CAL	FP,S; P,TC,RC	200,000	0.10	Aug. 1966
S-II Retro	~ 50	CAL	FP,S; P,TC,RC	391,000	0.10	April 1967
S-I, S-IB, S-IC Retro and S-II, S-IVB Ullage	2-3 ea.	CAL	SP	100,000-200,000	0.108,0.041, 0.063,0.103, 0.175	Aug. 1966
Liquid and Gaseous Engine Exhaust Tests						
F-1	~100	MSFC	S; P,T,TC,RC,R	Sea level	1/20	Oct. 1966
S-IC (5 F-1's)[3]	~ 50	MSFC	S,FP; P,T,TC,RC	Sea level	1/58	Summer 1967
J-2	~ 20	MSFC	S;P,TC,RC,A	Sea level	1/12.25	May 1967
S-IC(5 F-1's)[4]	~ 70	CAL	S,FP; P,TC,RC,SP	S.L. - 200,000	1/45	Summer 1967
J-2	2-3	AEDC	S; P,TC,RC,R,SP	100,000	Full	Summer 1967

[1] AEDC - Arnold Engineering Development Center, Tullahoma, Tennessee; OAL - Ordnance Aerophysics Laboratory, Daingerfield, Texas; RD./McG. - Rocketdyne, McGregor, Texas; MSFC - NASA/Marshall Space Flight Center, Redstone Arsenal, Alabama; CAL - Cornell Aeronautical Laboratory, Buffalo, New York.

[2] S = stagnation, FP = flat plate, P = pressure, T = temperature, TC = total calorimeter, RC = radiation calorimeter (wide view angle), R = radiometer (narrow view angle), SP = spectrometer, and A = ablation sensor.

[3] Also with solid propellant strap-ons.

[4] Also with solid propellant strap-ons and external flow.

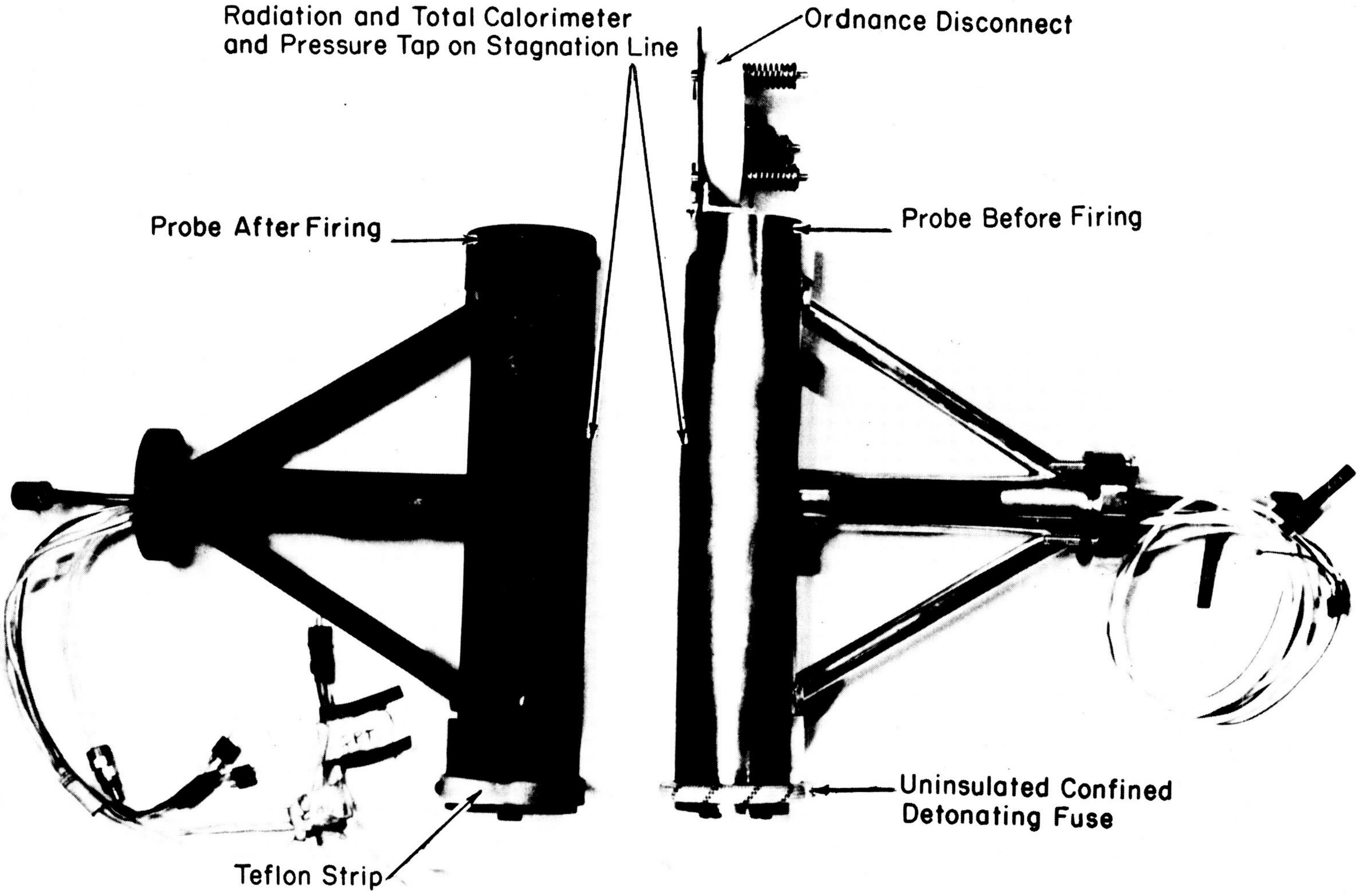

FIGURE I. COPPER HEAT TRANSFER PROBES USED IN S-II ULLAGE MOTOR TEST AT AEDC

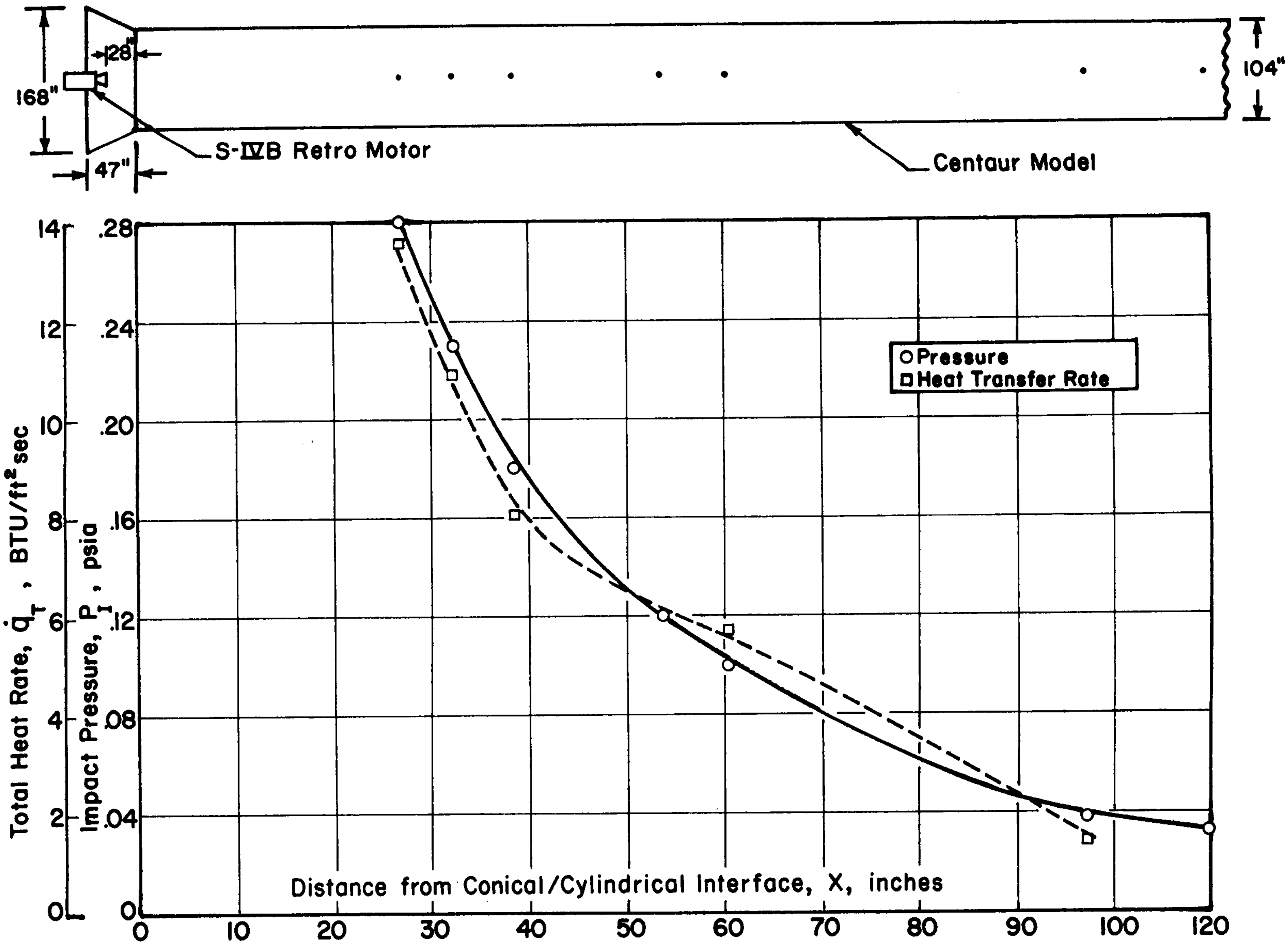

FIGURE 2. HEAT TRANSFER RATES AND PRESSURES ON CENTAUR PANEL DUE TO EXHAUST OF S-IVB RETRO MOTOR FIRED IN AEDC J-4 TEST CELL

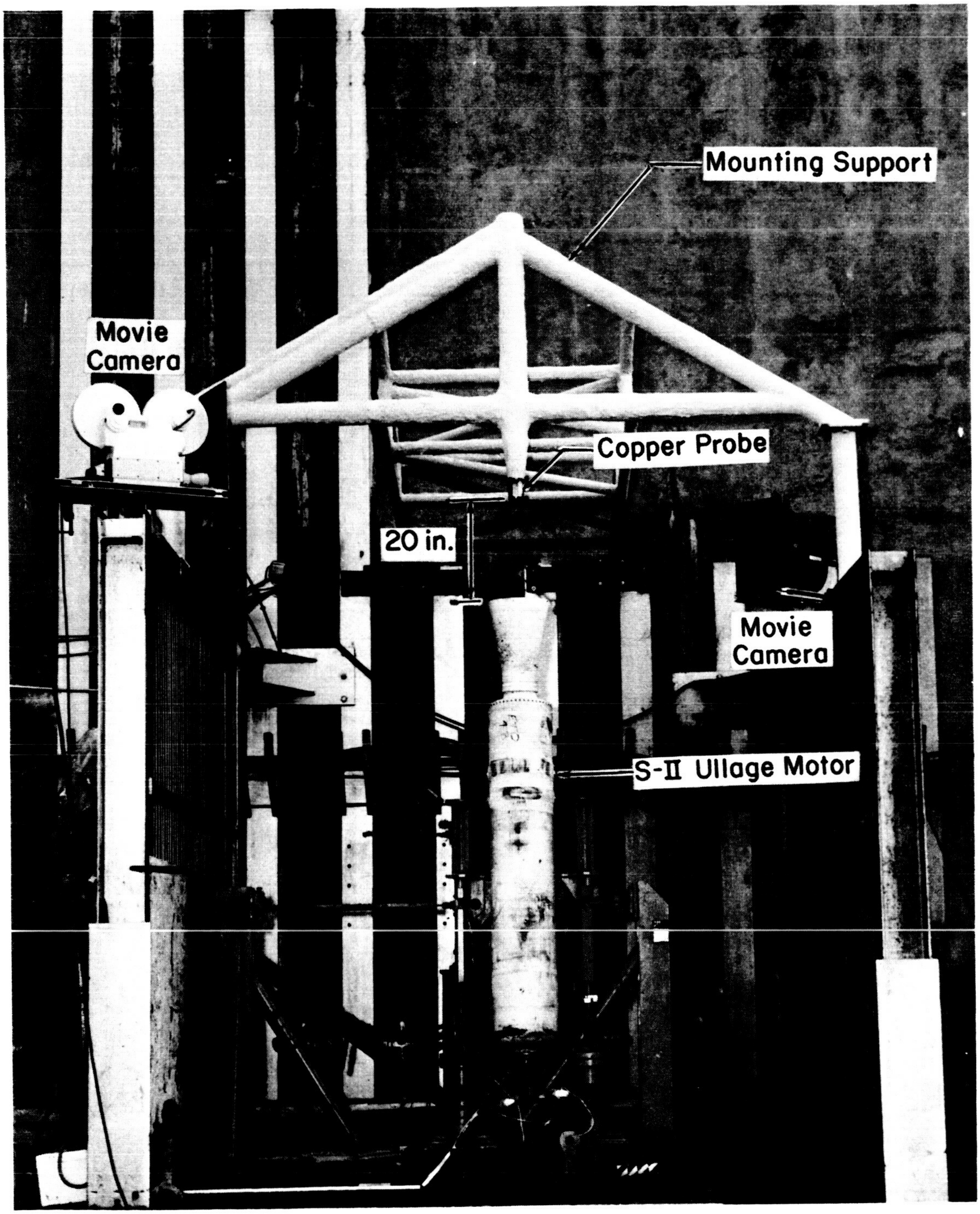

FIGURE 3. COPPER PROBE INSTALLATION FOR S-II ULLAGE MOTOR SEA LEVEL TESTS

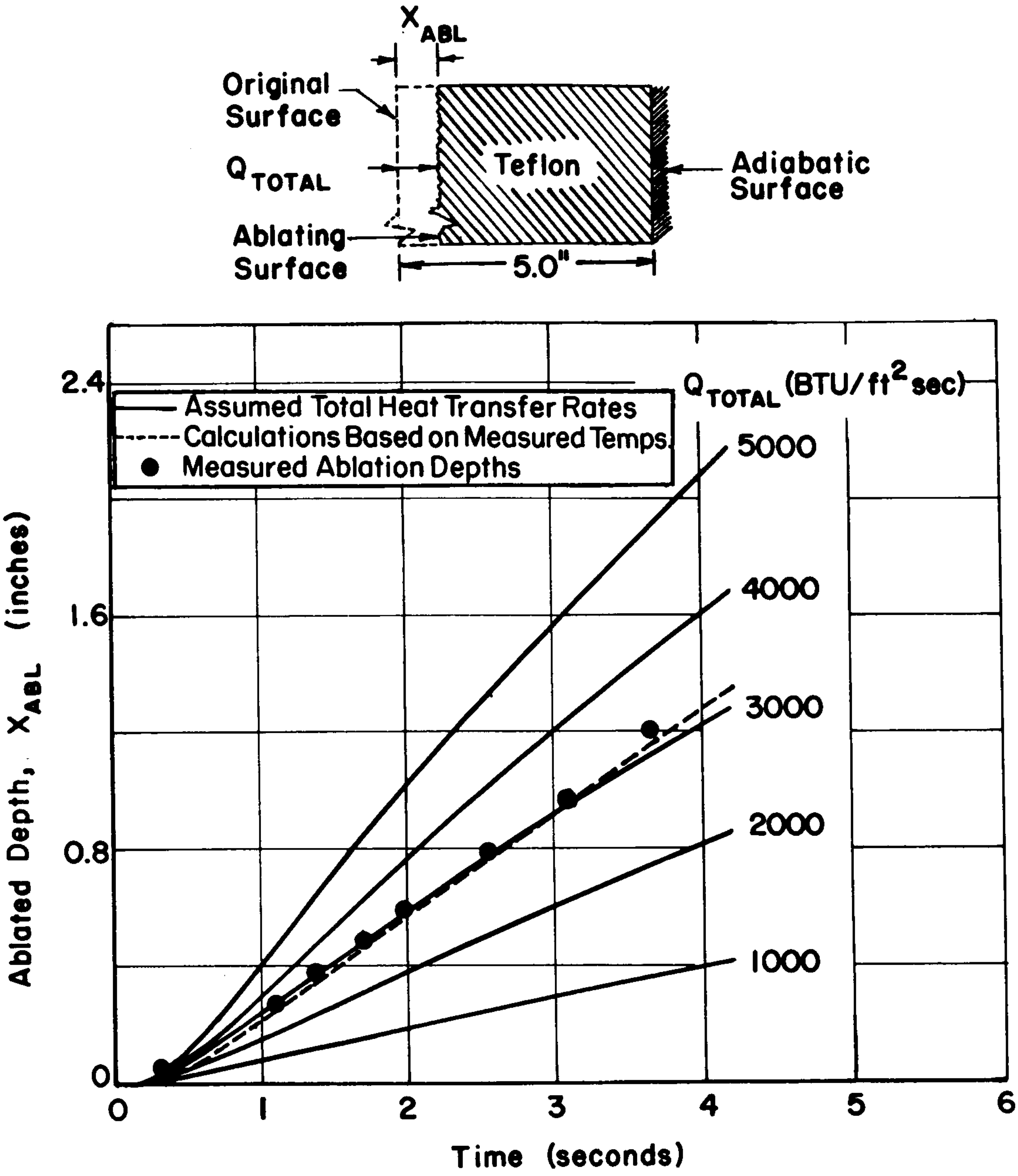

FIGURE 4. COMPARISON OF ABLATED DEPTHS IN TEFLON PROBE EXPOSED TO S-II ULLAGE MOTOR EXHAUST

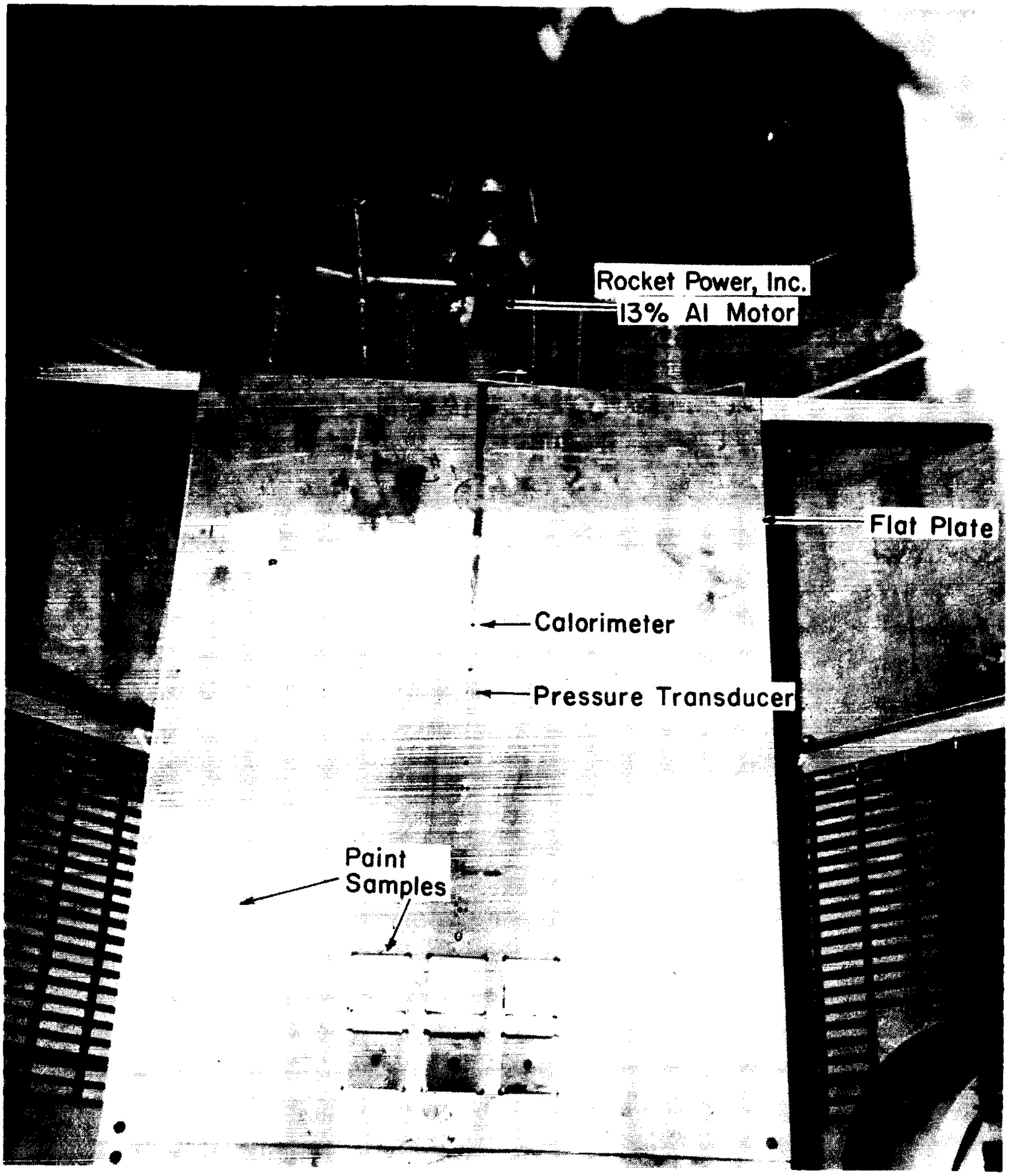

FIGURE 5. TYPICAL CENTAUR RETROROCKET INSTALLATION IN CELL 112 OF MSFC TEST LABORATORY

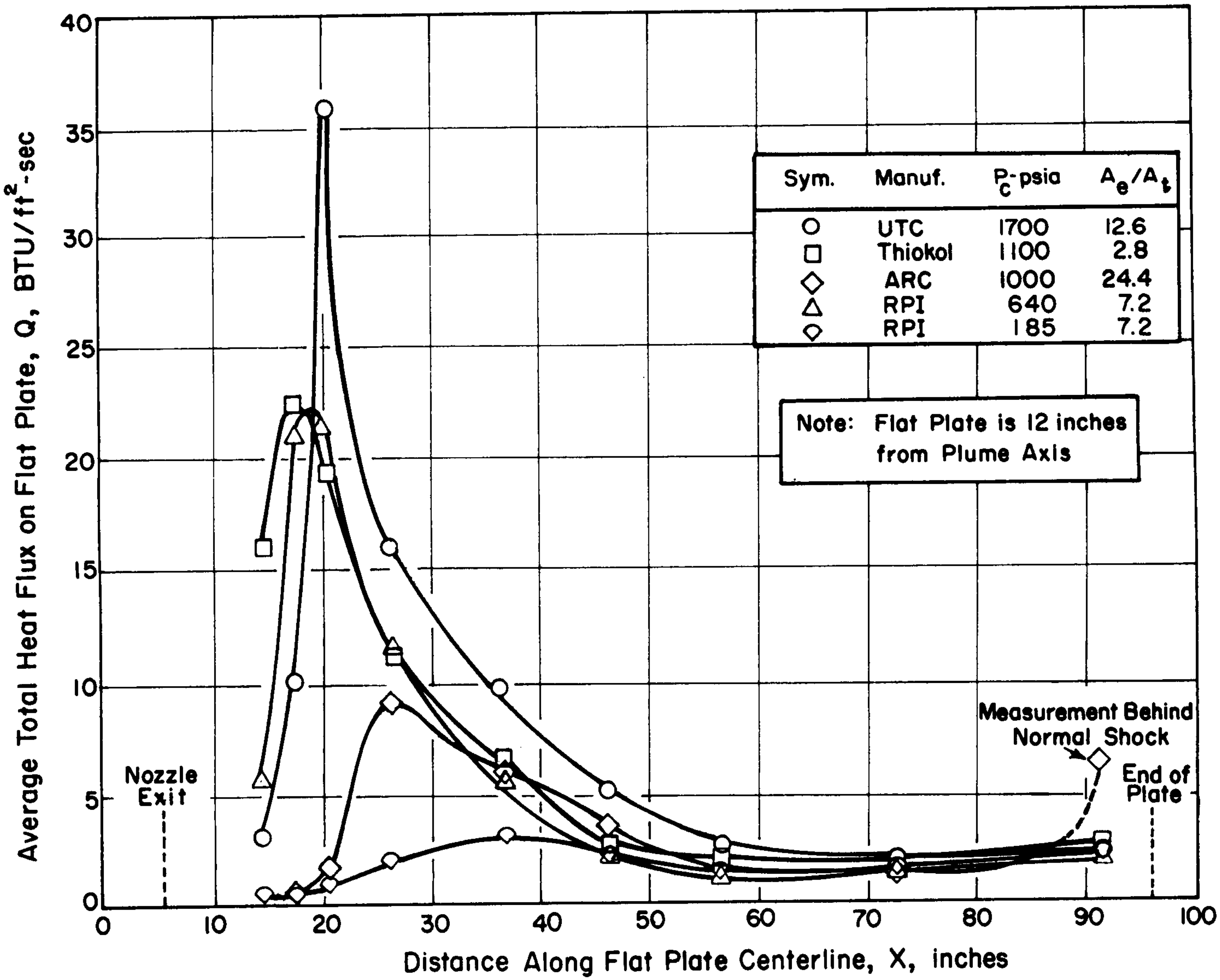

FIGURE 6. AVERAGE TOTAL HEAT FLUX ALONG FLAT PLATE CENTERLINE DUE TO CENTAUR RETRO ROCKET EXHAUST

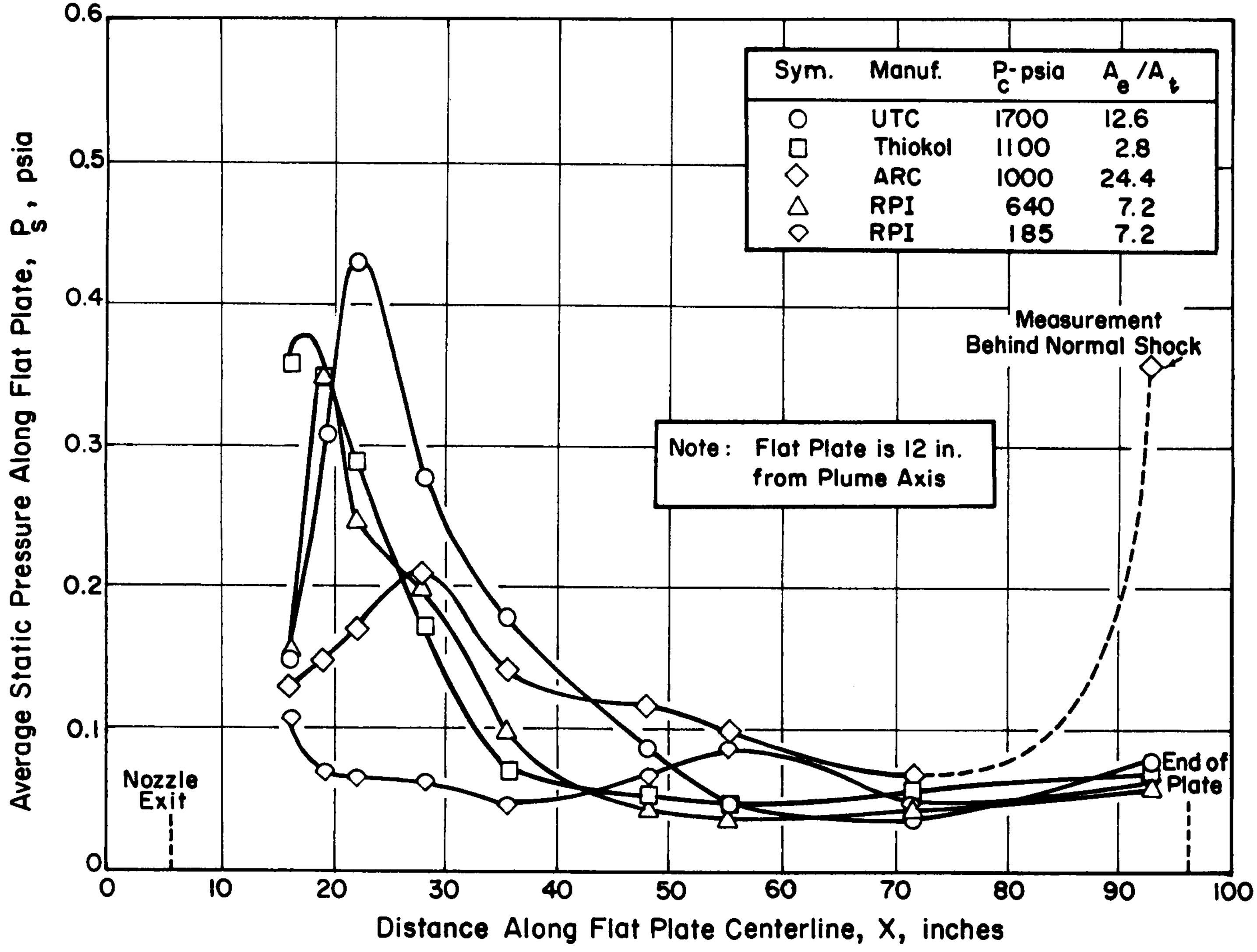

FIGURE 7. AVERAGE STATIC PRESSURE ALONG FLAT PLATE CENTERLINE DUE TO CENTAUR RETRO ROCKET EXHAUST

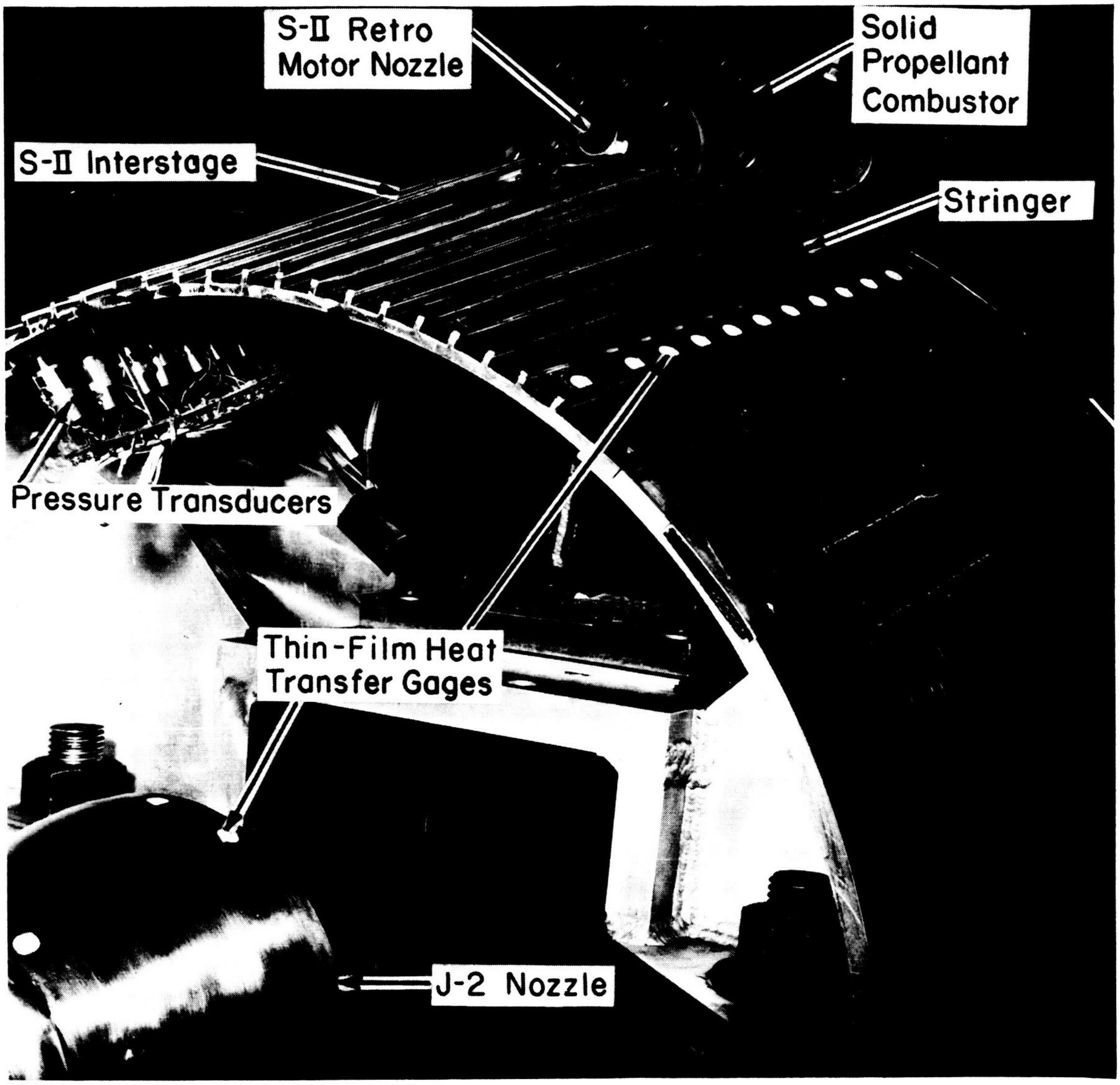

FIGURE 8. S-II INTERSTAGE MOUNTED IN CAL ALTITUDE CELL

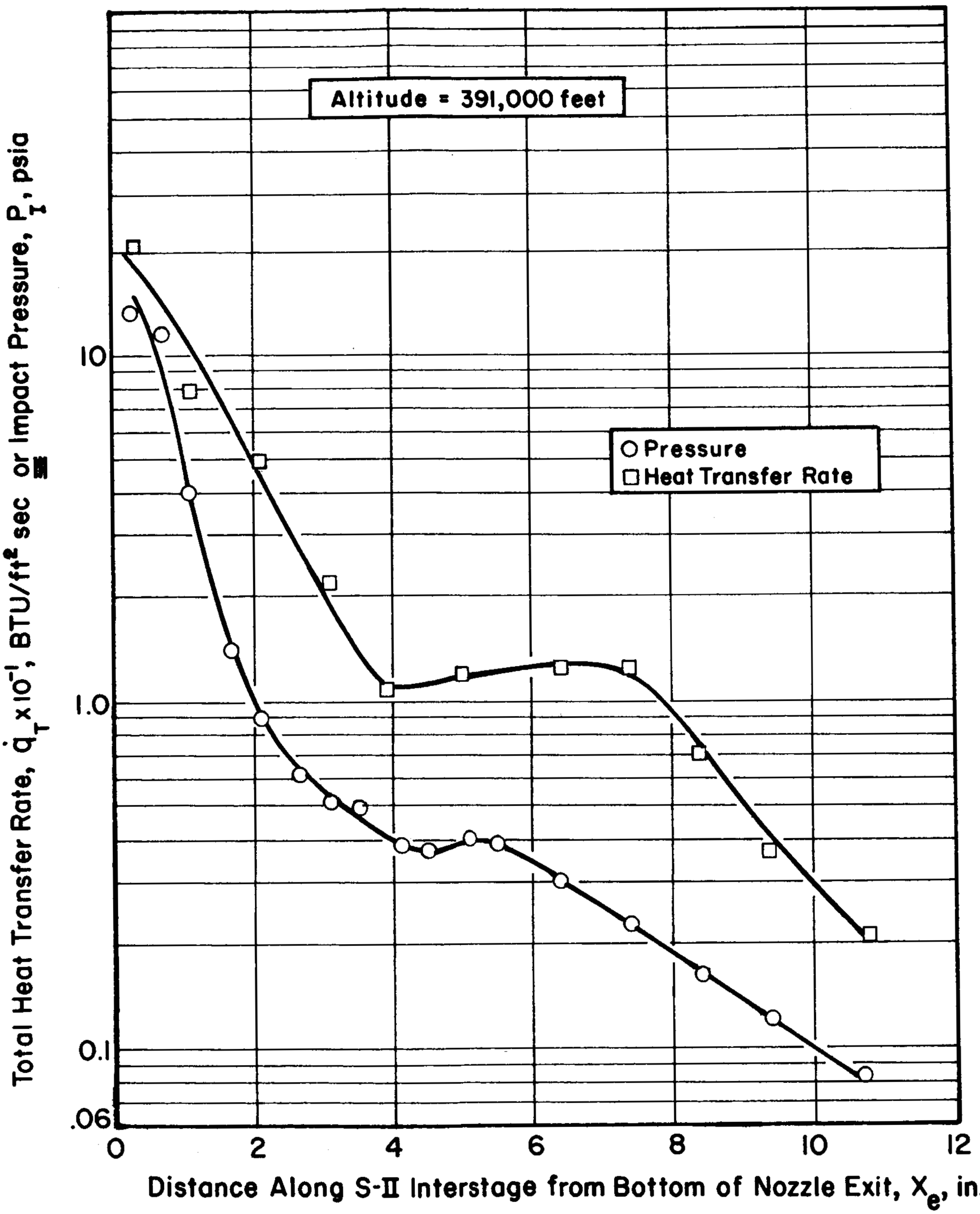

FIGURE 9. HEAT TRANSFER RATES AND PRESSURES ON S-II INTERSTAGE DUE TO EXHAUST OF S-II RETRO MOTORS FIRED IN CAL TEST CELL

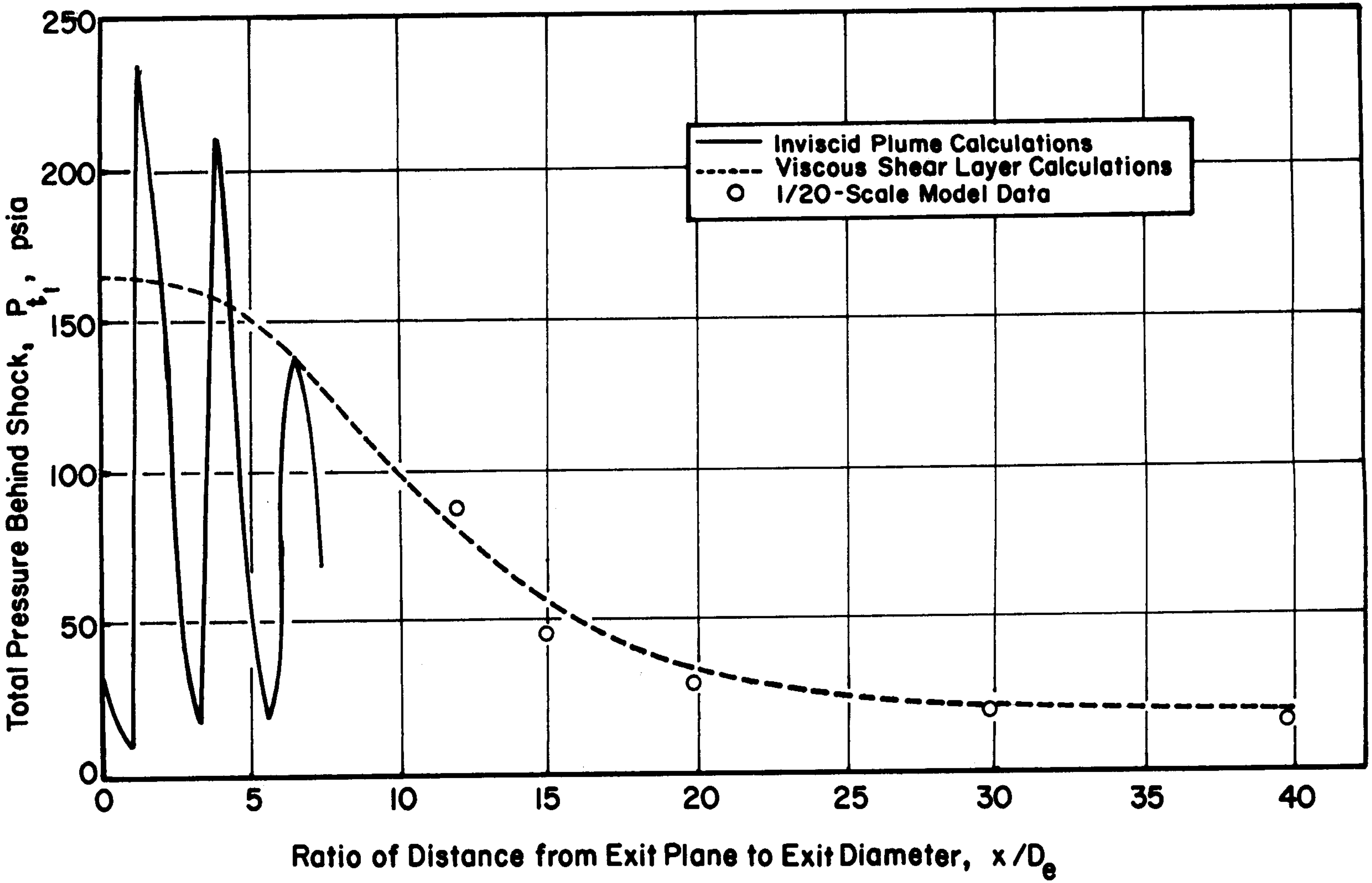

FIGURE 10. COMPARISON OF TOTAL PRESSURES BEHIND SHOCK (PITOT PRESSURE) ON F-1 ENGINE CENTERLINE

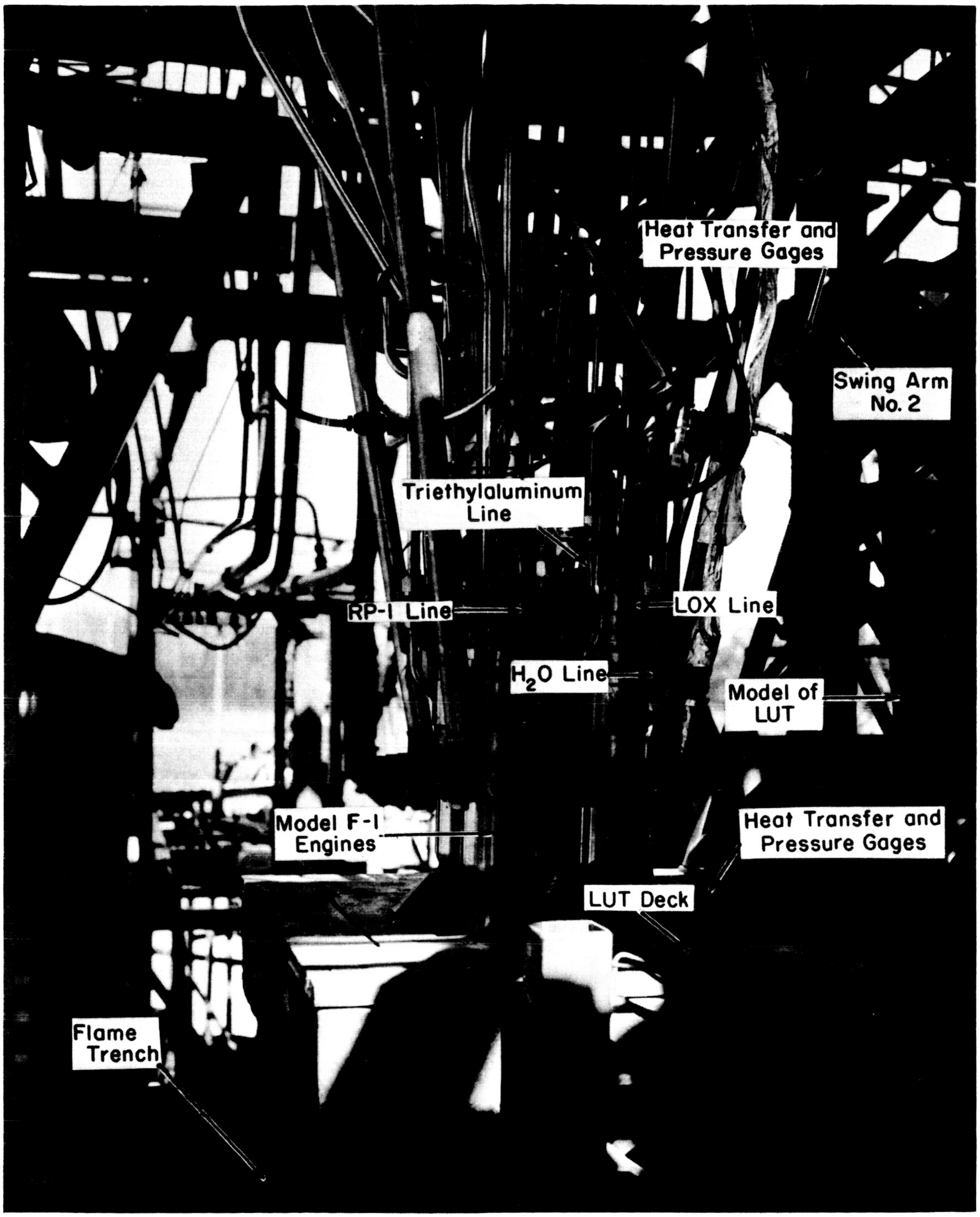

FIGURE 11. 1/58-SCALE MODEL OF S-IC STAGE, LUT, AND FLAME TRENCH AT MSFC TEST LAB

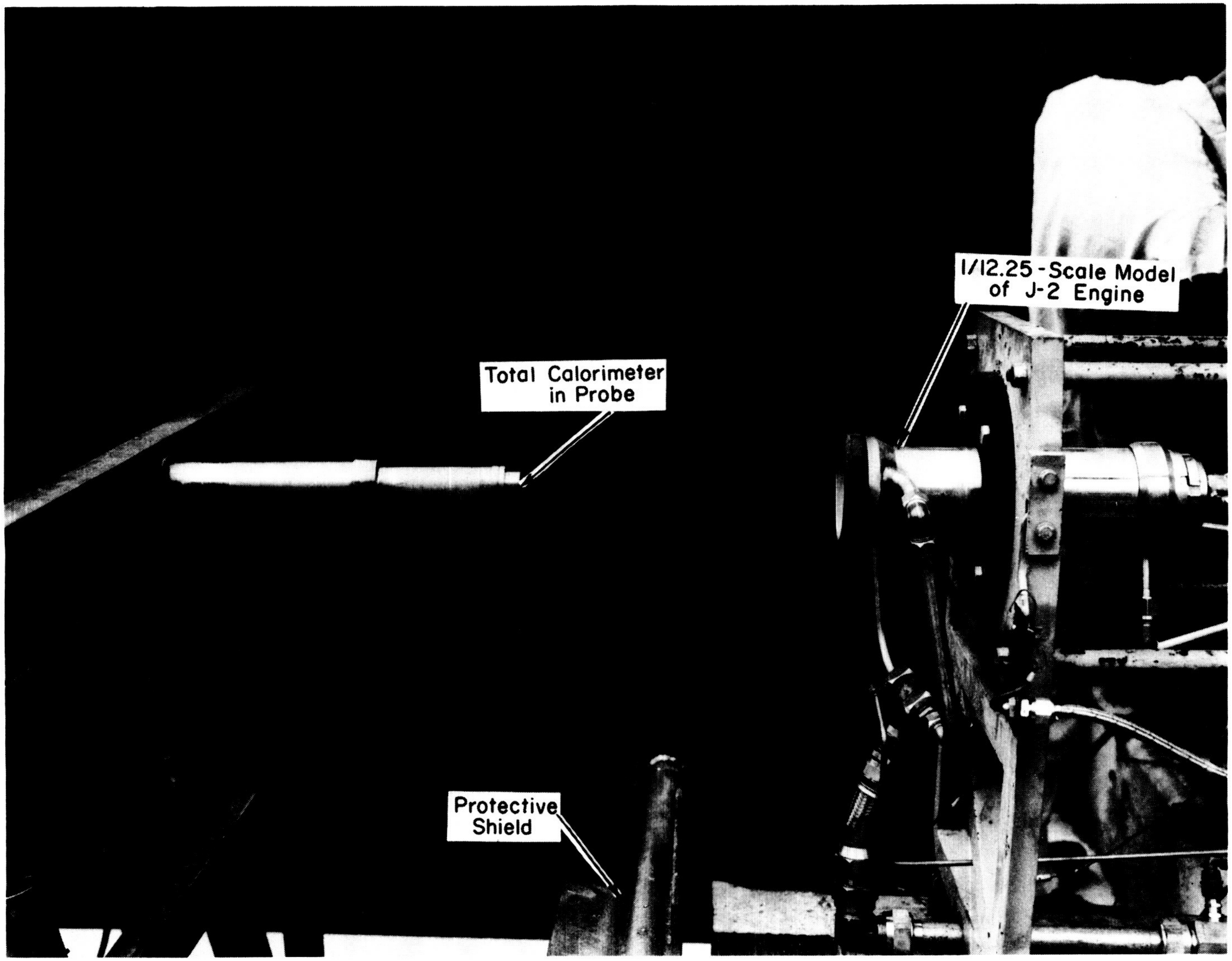

FIGURE 12. PROBE INSTALLATION FOR J-2 MODEL TESTS AT MSFC

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SUMMARY OF HEAT FLUX AND PRESSURE INSTRUMENTATION USED IN RECENT SATURN ROCKET EXHAUST TESTS

by William C. Rochelle

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

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